ABSTRACT

NEAR EARTH ASTEROID RENDEZVOUS (NEAR) ORBIT PHASE TRAJECTORY DESIGN

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Trajectory design of the orbit phase of the Near Earth Asteroid Rendezvous (NEAR) mission involves procedures that depart significantly from those procedures used for previous missions. On previous missions, the trajectory design involved finding a flight path that satisfied a rigid set of spacecraft and mission design constraints. A precise spacecraft ephemeris was designed well in advance of arrival at the target body. For NEAR, the uncertainty in the dynamic environment does not permit a precise spacecraft ephemeris to be defined in advance of arrival at Eros. The principal cause of this uncertainty is limited knowledge of the gravity field and rotational state of Eros. As a result, the concept for NEAR trajectory design is to define a number of rules for satisfying spacecraft, mission and science constraints and then apply these rules to various assumptions for the model of Eros. We thus have a nominal, high and low mass model of Eros that may be used for testing the trajectory design strategy and thus bracket the ranges of parameter variations that would be expected to include the actual parameters of Eros. The final design is completed after arrival at Eros and determination of the gravity field and rotational state.

Before defining a targeting strategy, it is necessary to define spacecraft and mission constraints that the spacecraft flight path must satisfy. These constraints are then transformed to trajectory design parameters and quantified. The final step in the design process is to target a trajectory that satisfies the numerical values assigned to these target parameters.

The spacecraft constraints that apply to Eros trajectory design include limits on fuel consumption, solar panel illumination and momentum wheel management. Other spacecraft constraints define the flexibility and speed that mission operations may be conducted. Probably the most important spacecraft constraint is to perform the prime mission within the allocated propellant budget. The propellant consumption constraint translates into about a 50 to 100 m/s delta velocity change during the orbit phase of the primary mission. This is a fairly generous allocation and should not be difficult to satisfy.

The most difficult spacecraft constraint to satisfy relates to solar panel illumination. Since the science instruments are fixed with respect to the spacecraft body, it is necessary to turn the spacecraft to point these instruments at Eros. In order to satisfy spacecraft power requirements, the solar panels cannot be turned more than about 30 degrees off the sun line. If the angle between the line to nadir and the plane perpendicular to the sun-line is greater than 30 degrees, the nadir point cannot be imaged without turning the spacecraft more than 30 degrees. A problem with this constraint is that the spacecraft is generally constrained to fly near the terminator during most of the mission.

Another important constraint relates to the time to conduct mission operations. In order to conduct the mission smoothly without resorting to round the clock operations, the minimum time between spacecraft propulsive maneuvers is limited to one week. The real limitation here is turning around accurate orbit determination solutions in time to perform the propulsive maneuvers that are required to keep the spacecraft on course. The differential velocity change resulting from maneuver execution errors corrupts the orbit solution. A rapid redetermination of the orbit places a large amount of pressure on the Mission Operations team to deliver accurate data and process this data into reliable solutions for the spacecraft orbit. By allowing a minimum of one week between maneuvers, this pressure is considerably reduced, since the amount of data available for the orbit solution is increased and the data quality is increased. A further benefit is more accurate orbit solutions in support of shape modeling and gravity field determination. The bottom line is that the more maneuvers that are performed, the more the orbit is corrupted and the more the quality of science is compromised. In addition, more risk to the mission is incurred because of poor trajectory control.

A trajectory design constraint related to orbit stability is that all low inclination orbits be retrograde with respect to the asteroid equator. Retrograde orbits are more stable because the faster relative motion of the spacecraft with respect to the asteroid tends to average out the effects of gravity harmonics. For this reason, synchronous prograde orbits are particularly unstable since the spacecraft lingers over the same point on the asteroids surface and may exchange enough energy to escape from or collide with the asteroid. In low orbit, even retrograde synchronous orbits may be unstable.

Science constraints on the trajectory design take the form of desires to obtain some particular orbital geometry and are generally not easily quantified. The requirement of the gamma ray spectrometer to obtain low orbits will drive the trajectory design through a series of orbits at various altitudes to hopefully satisfy all the science requirements on orbit geometry. The plan to stage the trajectory through a series of successively smaller circular orbits seems to satisfy imaging requirements, navigation requirements and makes the trajectory relatively simple to design. The general plan is to spend a specified amount of time in a series of circular orbits of pre determined radius. This keeps the mission on schedule and requires that only a single general imaging or mapping plan need be developed for any Eros gravity field that may be encountered. Transfer orbits between the circular orbits may also provide a unique opportunity for science observations from a perspective different from the circular orbits. However, the need to get to desired circular orbits may also make the transfer orbits unattractive for science.

The targeting strategy is simply an algorithm for translating the above spacecraft, mission and science constraints into a trajectory that can be navigated. The general approach is to develop a broad set of objectives and compute a series of propulsive maneuvers that will steer the spacecraft in the direction of satisfying these objectives. This differs substantially from the traditional approach of defining a number of constraints and searching for the trajectory that globally minimizes some performance criteria. The NEAR approach is to compute a maneuver that satisfies a local set of constraints and performance criteria and then propagate the trajectory into the future to see where it goes. At the appropriate time, a minimum of one week in the future, the constraints are reevaluated and another maneuver is computed. We continue in this fashion until all the science objectives are achieved. Of course we know in advance that we have enough propellant and time to make this simple strategy work.

In this paper the application of the NEAR trajectory design strategy to the current best estimate of Eros physical parameters is described. The resulting orbit design will be the prototype for the actual trajectory design to be carried out early in 1999. The trajectory is described and illustrated and some of the problems encountered in the design and there resolution is discussed.

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